1

Electrically Signaled Actuators (Signal-by-Wire)

The objective of this chapter is to present the architectures for the transmission/processing of information associated with actuation functions. The "flight control" application is used as the basis for this exploration because it is much more complex, and in many ways richer than other applications (e.g. landing gear, engines, etc.). This chapter will explain the evolution of flight controls from purely mechanical to all electrical signaling. To illustrate the importance of the electric transmission of information and power in modern aircraft, it is best to start off by stating a few orders of magnitude:

– the Airbus A380 has nearly 500 km of electric cables. The 120 miles of electrical cables for the Boeing B787 has a mass of 4 tons;

– Daniel [DAN 07] considers that on current long range aircraft, we can count 13,400 functional power lines (for both power and signal) for a total length of 240 km and with a total mass of 1,800 kg (we have to add 20% mass for connectors and fasteners). The part pertaining to the transmission of information is typically comprised of 7,800 lines, 170 km long and with a mass of 700 kg. This consists of 5,400 mono wire cables for analog and discrete signaling, with 2,050 twisted pair cables and 350 coaxial or quadraxial cables;

– for helicopters, [VAN 07] indicates that the Sikorsky S-92 is comprised of 295 kg of electrical cable, including 193 kg for information transmission and with 1,200 cable/aircraft interfaces. The mass of electrical cables increases to 612 kg on the CH-53K model, representing 3.5% of its empty weight.

1.1. Evolution towards SbW through the example of the flight controls

As indicated at the end of the previous chapter, the electrification of the information chain took place “gradually”. Whatever the application (fixed wing or rotor wing; commercial, military or private), when the electric information chain was first introduced, the mechanical information chain was maintained as a standby. The latter gradually disappeared, as redundancies applied to the power information chain allowed for the new alternative to reach the level of reliability required. The richness of this evolution, as illustrated by flight control, is analyzed below. It also attempts to embrace other actuating functions associated with the landing gear (steering and braking) or the engine (control of air and fuel flows, geometry and thrust reversers).

1.1.1. *Military applications*

The trends towards the uptake of more electrical information chains were initially driven by the need to autopilot flight in situations with zero visibility (military aircraft) or without an actual pilot (missiles and space launchers). For military aircraft, reducing vulnerability and increasing maneuverability have also been major reasons for the electrification of both the information and the control chains. This is illustrated in Table 1.1, which has mainly been sourced from [RAY 93].

Year	*Manufacturer model*	*Advancements (from the previous model)*
1943	Boeing B17E	3-axis autopilot with electrical output, mechanical summation on pilot commands by an electro-mechanical actuator
1959* 1964**	Dassault Mirage IV	Electrohydraulic elevons, analog electric signaling with mechanical backup
1964* 1967**	General dynamics F-111	Electrohydraulic spoilers
1974* 1978**	General dynamics F-16	Analog quadruplex FbW to increase relaxed static stability Quadruplex electronics with middle value logic Side-stick
1974* 1982**	Panavia Tornado	Triplex digital electronic FbW and mechanical backup channel

1978* 1983**	McDonnel Douglas F/A-18	Quadruplex digital electronic FbW Separation of electrical and hydromechanical fault detection and isolation Backup mechanical signaling for pitch control
1974* 1986**	Rockwell B1-B	Combination of full authority SCAS and purely hydromechanical control chains Structural Mode Control System (or SMCS)
1979* 1984**	Dassault Mirage 2000	Electrohydraulic actuators, and quadruplex analog FbW, electrical backup
1986* 2001**	Dassault Rafale	Electrohydraulic actuators, triplex digital FbW and analog backup Side-stick
1989* 1997**	Northrop B2	Quadruplex digital electronic FbW 4 Actuator Remote Terminal (or ART) in the wings, communication by multiplex bus Performance level kept in case of failure of an actuator electronic channel by changing the loop control gains of remaining healthy channels

*First flight, **Entry into Service (EIS)

Table 1.1. *Evolution towards an all-electric information chain for the flight controls of military aircraft*

1.1.2. *Commercial aircraft*

Table 1.2 shows the incremental evolution of flight controls for commercial aircraft, using European aircraft as an example. Note that it took a cumulative total of over 40 years to completely replace the mechanical information chain with all-electric information chain.

Year	*Manufacturer model*	*Advancement (from the previous model)*
1969* 1975**	Sud Aviation - British Aircraft Corporation/ Concorde	Analog FbW with mechanical backup Analog electrical signaling between cockpit and actuators
1972* 1974**	Airbus/A300B	Analog FbW for 12 non-essential functions Analog Electrical signaling between cockpit and actuators Position servo control performed by computer in the cockpit

1982* 1983**	Airbus/A310	24 FbW actuators controlled by five computers Removal of low speed ailerons (roll control assisted by spoilers at low speed) Introduction of electrically signaled trim for ailerons and rudder
1987* 1988**	Airbus/A320	FbW on 3-axis, 7 digital computers Mechanical signaling between the yaw damper and the rudder actuators Backup mechanical signaling (pseudo FbW) for rudder and Trim Horizontal Stabilizer (THS) Introduction of the side-stick
1991* 1993**	Airbus/A330 340	5 digital computers Removal of any mechanical signaling for the rudder control (A340-600) with resort to an electrical analog Backup Control Module (or BCM)
2005* 2007**	Airbus/A380	6 digital computers Removal of the last mechanical backup signaling for THS control (*Full* FbW)
2013* 2015**	Airbus/A350	6 digital computers Position servo-control at actuators level (Actuator Control Electronics or ACE)

*First flight, **Entry into Service (EIS)

Table 1.2. *Evolution towards an all-electric information chain for the flight controls of European commercial aircraft*

The advances in information chains that are mentioned in this table resulted in a significant and continuous improvement on performance [VAN 02]. Of particular note:

– between the A300-B4 and the A310, there was a mass reduction for flight controls of approximately 300 kg, predominantly due to the removal of low speed ailerons permitted by the complementary use of spoilers for low speed roll control;

– between the A310 and the A320, a mass of 200 kg through improved flight controls, an increase in security by the protection of the flight envelope, a reduction of the pilot load that negated the need for a flight engineer (two-crew operation) and a reduction of dynamic airloads through the Load Alleviation Function (or LAF);

– between the A300 and the A340, a 45% mass reduction of the rudder actuators (of identical specifications) and a further mass reduction of about 50 kg on the A340-500/600 from the removal of the mechanical controls and the associated actuators for the yaw damper, in favor of an electric control (see Figure 1.20);

– on the A380, a mass saving through reduced stability margins for longitudinal balance (area of the trim horizontal stabilizer reduced by 10% thanks to the introduction of FbW for pitch control, including for the backup channel);

– on the A350 [AIR 13], an increase in the aerodynamic efficiency of the wing due to the Differential Flap Setting (or DFS), the Variable Camber (or VC) and the Adaptive Dropped Hinge Flaps (or ADHF).

1.1.3. *Helicopters and compound helicopters*

On airplanes, lift is provided to the wing due to relative airspeed. Furthermore, a given attitude control function (at least for roll or pitch) is often provided by several moving surfaces, themselves driven by a redundant actuation system (e.g. two separate actuators as on commercial airplanes or a dual body actuator as with many military aircraft). Given the dynamics involved, it is generally accepted to temporarily lose the positioning function of a control surface in the event of actuator failure. We can even accept to permanently lose the positioning function, provided that actuators respond as required to a failure (depending on the need: fail-passive, fail-neutral or fail-freeze; see Volume 1, Chapter 2).

On helicopters, lift and attitude are controlled by the cyclic pitch and collective pitch of the blades. The angle of attack of each blade must be controlled absolutely at each moment: as such any transient loss of the positioning function is not permitted[1]. Moreover, in the event of a failure of the flight control functions, the boundaries of the flight envelope are reached significantly faster than on a plane.

1 For example, we typically tolerate a runaway at maximum speed during of 20 ms or less, which produces a displacement of rotor actuator rod not exceeding 5 mm.

All these considerations explain why the transition of flight control signaling, from mechanical to electrical, takes so long to occur. This is illustrated in Table 1.3 which indicates that the development of helicopter flight controls has taken nearly 70 years [STI 04]. Faced with the criticality of helicopter flight controls, FbW was gradually phased in but commands remained transmitted mechanically. At any time, this design enabled a safety pilot to take the control through a backup mechanically signaled channel in case of failure of the FbW channel.

Year	*Manufacturer model*	*Advancements (from the previous model)*
	Increase in mechanical stability and dynamics	
1941	Young patent [YOU 45]	Hover stabilization by Young-Bell bar
1953	Sikorsky/H03-S1	Improved longitudinal stability through action on the blades pitch as a function of aerodynamic forces generated on air foils
1954	Bell/47	Additional vertical acceleration feedback by inert mass
1964	Bell/H13	Mechanical first-order lead filter to improve the roll rate response
	First electronic autopilot	
1950	Piasecki/HUP-1	Introduction of the electronic autopilot
1952	Sikorsky/S-56	Improved response to gusts and controllability
1960	Sikorsky/S-58	”Hands-free” hovering
	Research in the United States to improve reliability and maturity	
1973	Boeing/CH47B	Demonstration of digital FbW with redundancy management (TAGS project)
1975	Boeing/XCH-62	Response to commands and stability augmentation level made dependent on the phase of flight (HLH project)
1986	Sikorsky/UH-60A	Optic/electric flight control with evaluation of side-sticks and improved agility (ADOCS project)
1992	Sikorsky/ S-76 Shadow	Transition during the takeoff and landing phases
	Research in Europe and Canada	
1979	Bell/205	FbW demonstration flight, NRC, Canada
1985	BO/105 ATTHeS	FbW flight simulation, DLR, Germany

1991	Aérospatiale/ AS 365 Dauphin	FbW demonstration flight, Aérospatiale, France
2001	Bell/412 ASRA	FbW demonstration flight, NRC, Canada
2002	Eurocopter/EC135	Fly-by-light demonstration flight, DLR, Germany

Table 1.3. *History of the development of electric flight controls for helicopters and compound helicopters*

Table 1.4 confirms the difficulty and time spent for moving FbW designs into service: in 2015, the only models with FbW in mass production were the NH Industries NH90 and the Boeing V22 Osprey; while several programs have since been abandoned.

Year	*Manufacturer model*	*Characteristics*
1975[*] 1984[**]	Boeing/ AH-64A Apache	Attack Helicopter Mechanical signaling with backup FbW on each axis (*Back-Up Control System* or BUCS)
1989[*] 2005[**]	Boeing/ V22 Osprey	Multi-role combat tiltrotor
1992[*] 1994[**]	Mc Donnel/ MD 900 Explorer	Light helicopter Analog FbW for vertical stabilizer (Vertical Stabilizer Control System or VSCS)
1995[*1] 2003[*2] 2007[**]	NH Industries/ NH-90	Middle class military helicopter First mass-produced military helicopter equipped with FbW (quadruplex) series
1996[*] Abandoned 2004	Boeing/Sikorsky RAH-66 Comanche	Combat Helicopter, FbW (triplex) without mechanical backup
1998[*] Abandoned 2014	Sikorsky/CH-148	Multi-role naval helicopter
2003[*] 2018[**] ?	Agusta Westland/ AW609	Civil transport tiltrotor equipped with FbW (triplex)
2015[*] 2017[**] ?	Bell/ 525 Relentless	Multi-role civil transport helicopter. First commercial helicopter equipped with FbW (triplex) without mechanical backup.

[*] First flight, [**] Entry Into Service (EIS), [1] mechanical mode only, [2] full FbW mode

Table 1.4. *Examples of FbW industrial helicopter and compound helicopter programs*

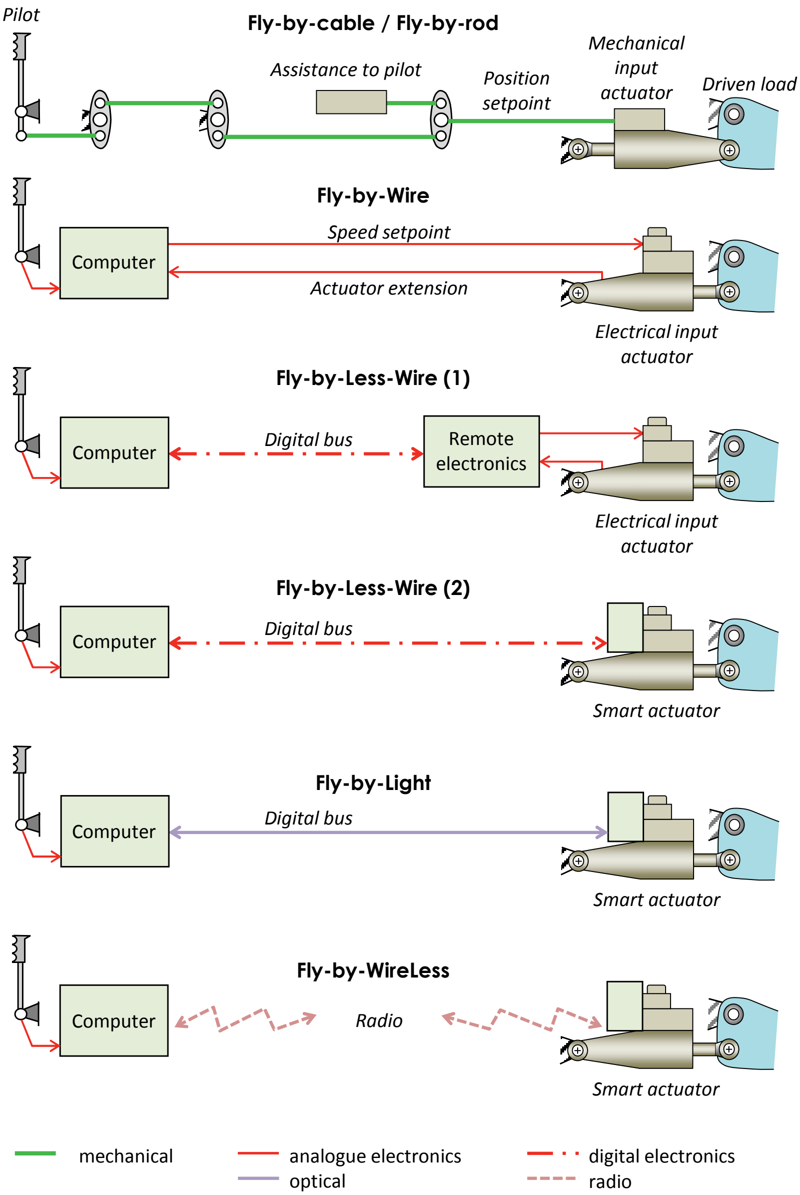

Figure 1.1. *Major developments of flight control signaling*

1.2. Incremental evolution from all mechanical to all electrical

Under the signal aspect, the evolution of flight control can be shown in Figure 1.1. Each architecture, presented here in a simplified way, will be discussed in further detail in the following sections. In practice these changes occurred in small increments, which are not all shown by this figure:

– there were no actuators on the first flight controls;

– between the Fly-by-Cable and Fly-by-Wire, we could have distinguished the introduction of actuators with dual inputs, both electrical and mechanical. First, the electrical input appeared alongside the servovalve, to directly inject into the actuator the augmentation or autopilot commands. Next, the electrical input became the main input, with the mechanical transmission path being used as a backup before being gradually phased out.

1.2.1. *Exclusively mechanical signaling*

In the designs of conventional flight controls, the pilot's commands move the load exclusively transmitted in mechanical form: here we might speak of "Fly-by-Cable" or "Fly-by-Rod" versus "Fly-by-Wire". Figures 1.2–1.4 provide examples of such architecture as employed in military aircraft (McDonnell Douglas F-15, EIS 1976), commercial aircraft (Airbus A310, EIS 1983) and helicopters (Sikorsky S-76, EIS 1979).

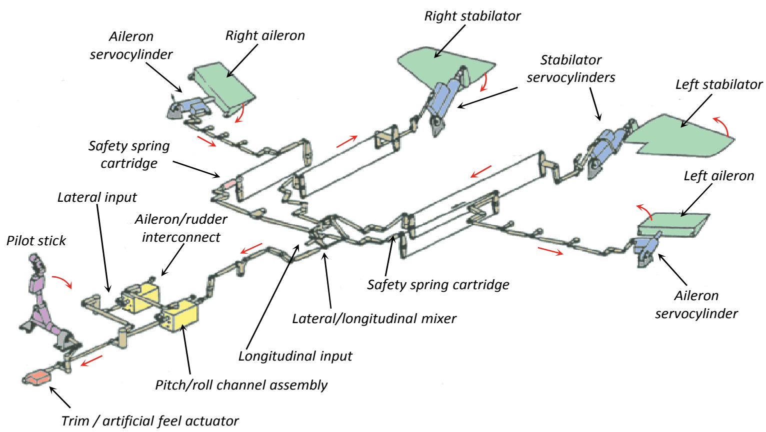

Figure 1.2. *Roll control for the McDonnell Douglas F-15 fighter (http://www.f15sim.com). For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

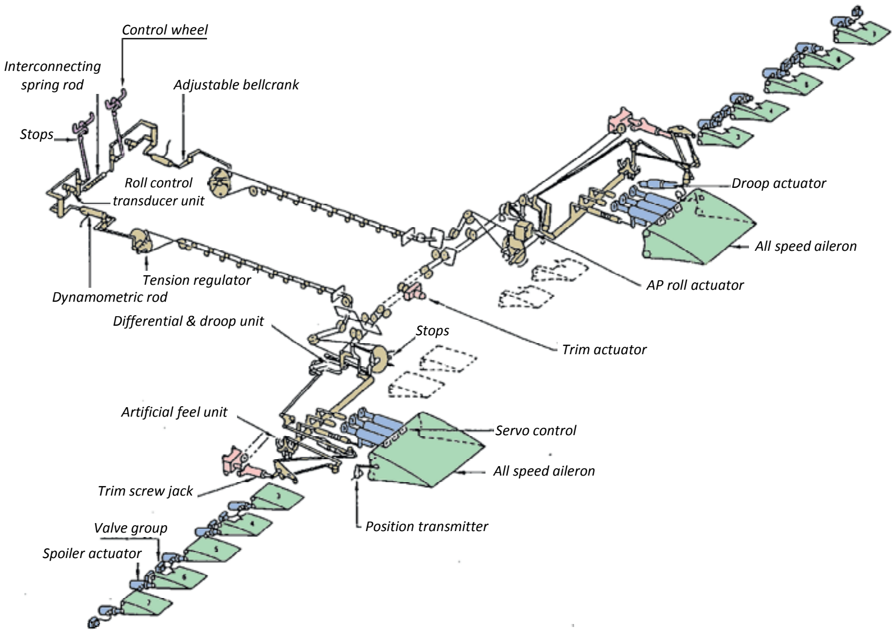

Figure 1.3. *Roll control for the Airbus A310 [VAN 02]. For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

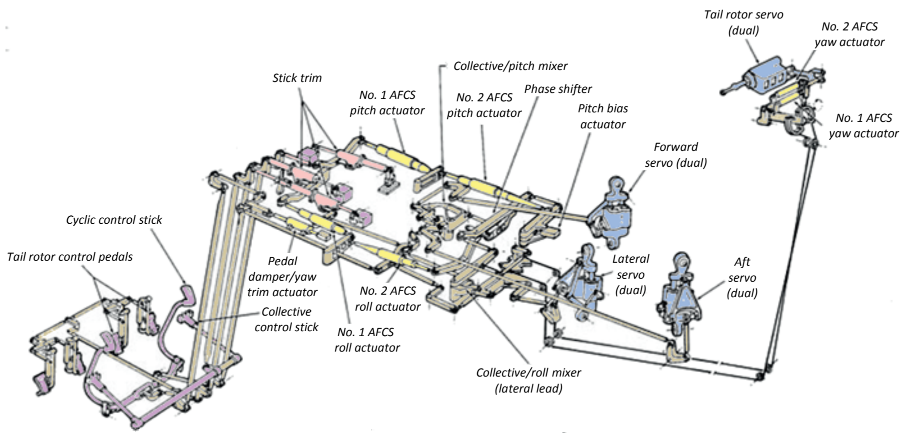

Figure 1.4. *Flight controls of the Sikorsky S-76 helicopter (© I. Sikorsky historical archives). For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

While different aircraft types have different strengths, their architectures share many similarities, particularly those listed functions and their gradual introduction. The addition of these functions is realized by mechanical summation through two main principles:

– The position (or movement) summing. For the example given in Figures 1.5(a) and (c), this is performed by crossbar, or hydraulic or electro-mechanical actuators integrated in series. That is to say, each of these channels adds a displacement on the mechanical signal chain. A feared event would be a break in one of these channels, as in this case the remaining mechanical parts can move independently of one another;

– The force summation. As shown in Figure 1.5(b), it is realized by an arm whose position is given by one input and propagated to others. A feared event would be the seizure of a pathway, thereby freezing the entire mechanical chain. To avoid this situation, we usually install spring rods, Figure 1.5(d), on the chains in series, so as to decouple the channels, thereby overcoming force. Note the example in Figure 1.3, at the interconnection of the two pilots' control sticks, or again for the transmission of mechanical position setpoints to the three aileron hydraulic cylinders from a single source movement.

1.2.1.1. *Primary piloting functions*

The pilot uses conventional inceptors, control stick/control wheel and rudder pedals to change the position of the loads to be actuated (in these examples, flaps, swashplate of the main rotor and/or the tail rotor).

The pilot's orders are transmitted mechanically to the loads to be actuated through connecting rods, cables, levers, cranks or even flexible cables (ball bearing cables) as employed in the McDonnell Douglas F-15 Eagle fighter. When needed, further elements are installed to ensure tension regulation if the transmission is achieved by cable. Figure 1.6 shows the flight control cables from the Aero Spacelines Super Guppy cargo aircraft. For loading and unloading, the hinged nose of the plane that constitutes the cargo door is pivoted about a vertical axis. Thereby it requires all the flight control cables to be disconnected.

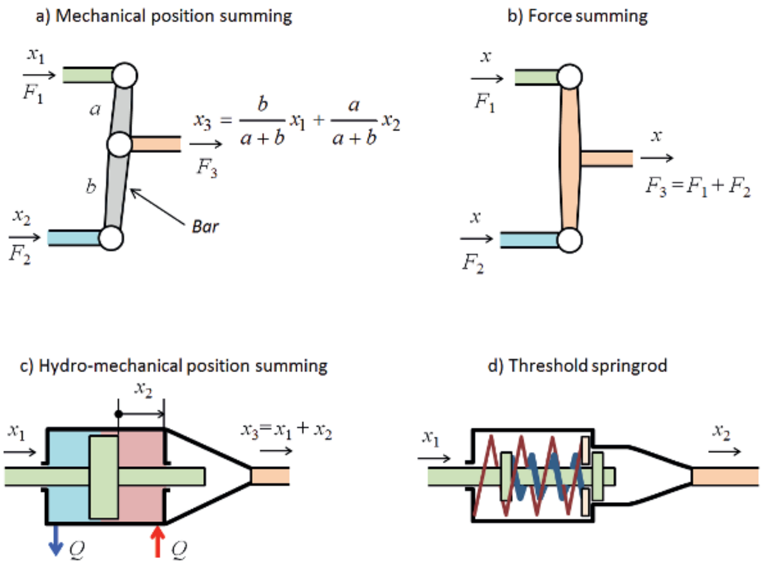

Figure 1.5. *Summation principles of mechanical transmission. For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

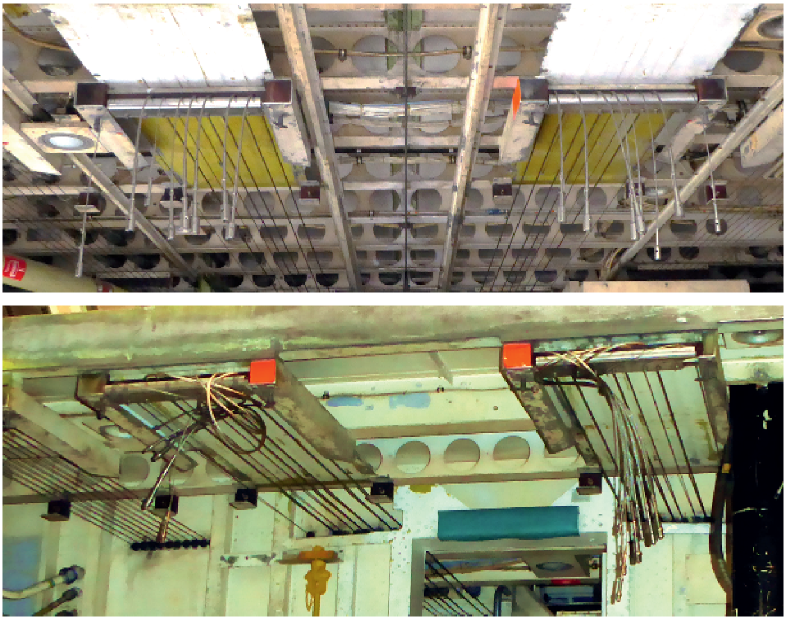

Figure 1.6. *Flight control cables of the Super Guppy cargo plane*

If needed, the mechanical commands are combined in a mechanical and passive manner, using the position-summing concept. Thus, coupling or decoupling functions are produced. Figure 1.3 shows an example of droop actuators that steer the ailerons down to improve aerodynamic efficiency when the flaps are extended. Figure 1.4 provides a further example through the mixer that develops swashplate commands, based on the collective and cyclic orders generated by the pilot.

1.2.1.2. *Augmentation and automation of control commands*

Functions for the augmentation and automation of control commands are added through the auxiliary actuators operating on the mechanical information chain. As shown in Figures 1.2–1.4:

– AutoPilot (AP) or Automatic Flight Control System (AFCS) actuators;

– compensation actuators (trim) that balance the aircraft for a particular flight phase when pilot commands are at neutral;

– actuators for specific functions such as Stability Augmentation System (or SAS). For example, the yaw damper on the yaw control command to avoid the Dutch roll;

– actuators for limiting the structural loads such as the rudder travel limiter as a function of true airspeed;

– actuators for specific functions such as the Control Augmentation System (CAS) in manual control.

Figure 1.7 shows how the auxiliary actuators are integrated into the mechanical signal chain for a conventional flight control axis[2]. It is represented in the generic intermediate form, addressing both aspects of signal and power.

2 As shown in Figure 1.8, of Volume 1, the first functions (e.g. primary flight controls) of this type were initially implemented without any electrical device.

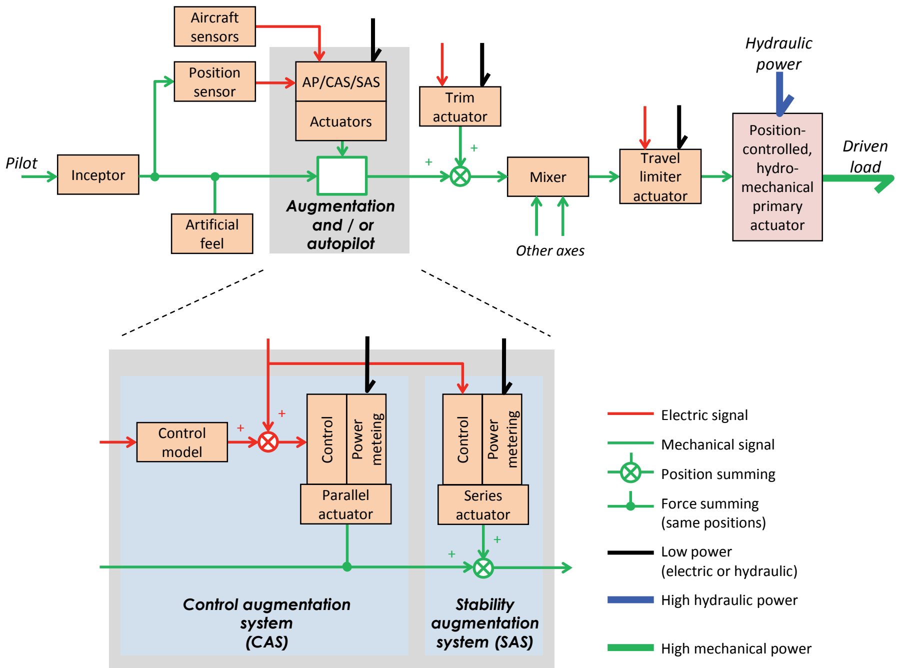

Figure 1.7. *Generic architecture, signal and power, for a conventional flight control axis. For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

The bottom half of the diagram shows an example for the implementation of an augmented or autopilot system. In practice, this implementation is slightly different depending on the type of aircraft (fixed or rotary wing aircraft):

1) Stability Augmentation System function. The SAS actuator is mounted geometrically in series on the mechanical signal chain to produce a summation of position. It must be quick enough to respond to aerodynamic disturbances. Its total stroke is only a few percent of that of the mechanical chain, which allows for manual control in the event of SAS malfunctions. It is located downstream of the artificial feel system. Therefore, it produces orders that propagate towards the main actuator, which do not alter the position of the pilot inceptors: its action is thus invisible to the pilot who does not perceive it.

2) Control Augmentation System function. The CAS actuator is geometrically mounted in parallel: it acts by imposing the position command

relative to the airframe to the primary actuator. Its action is thus directly reflected at the level of pilot inceptors and consequently perceived by the pilot. The actuator has a higher authority, several tens of percent, of the functional stroke of the mechanical chain. In the event of the function failing, the pilot must be able to counter the forces produced by the actuator which should therefore include a release system. Conversely, in the case of a malfunction of the mechanical drive chain upstream of the actuator, the actuator can be used as a backup to pilot the aircraft via the commands issued by the autopilot. The force summation that can be seen as position sharing is implemented in most cases, as shown in Figures 1.2–1.4, by means of spring rods.

3) Compensation function. The trim actuator performs an average compensation that only requires a very low bandwidth. It is realized through position summing on the mechanical chain so as to ensure that the aircraft is aerodynamically balanced. This reduces the average forces that the pilot needs to generate and maintains the inceptors at neutral position, regardless of the stabilized flight phase. The trim actuator must be disconnected in the event of malfunction.

4) Load limiting function. The travel limit actuator bounds the permitted stroke according to flight conditions. This ensures the integrity of the structure by ensuring that the aerodynamic loads do not exceed the limit values used for sizing.

The power that must be generated by auxiliary actuators is generally low (a few tens of N, a few cm/s). On the other hand, their criticality is often moderated because they can be disengaged or surpassed by the pilot. Hence why it is possible to achieve this in electro-mechanical form, as is generally the case for commercial aircraft. When they need to be more dynamic, these actuators are usually electrohydraulic (see Volume 1): the hydraulic power metering function is performed by an Electro-Hydraulic Servo Valve (EHSV) or a Direct Drive Valve (DDV). In addition, the hydraulic technology can likewise realize a compact and easy way to accommodate these disengagement functions. This type of actuator is found on helicopters for the AFCS or SCAS functions. Furthermore, the CAS and SAS functions can be integrated as a single function (SCAS, AP or AFCS) using a single actuator mounted in series on the information chain. This function can be made redundant. For helicopters, integration into the airframe is facilitated by grouping these auxiliary actuators on a single physical unit in charge of the three (or four) axes of motion: roll, pitch, yaw (and collective pitch), as shown in Figure 1.8 for the Eurocopter

Tiger SCAS. Note the presence of the dual input servovalve, mechanical (position control) and electrical (SCAS function).

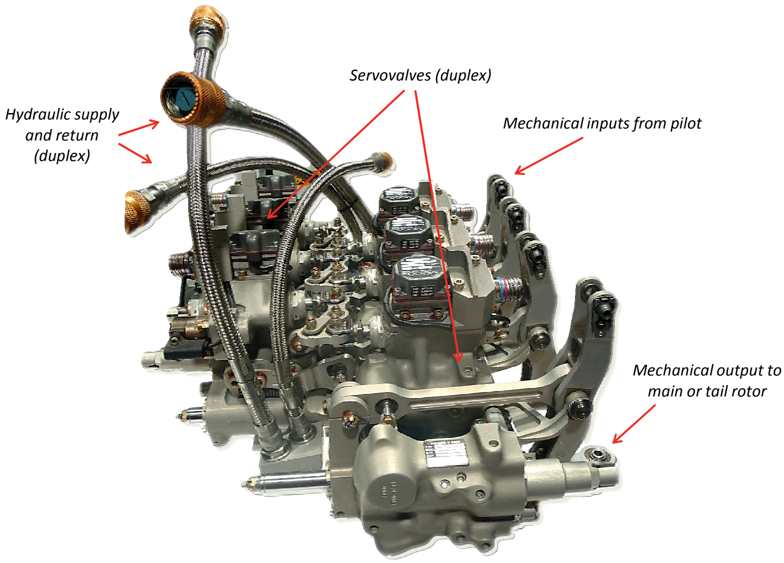

Figure 1.8. *Auxiliary actuator SCAS group of the Eurocopter Tiger helicopter*

Figure 1.9 shows the CAS servovalve control diagram for the pitch axis control of the McDonnel Douglas F-15 fighter. Note the large number of inputs and analog functions to be performed electronically to implement the CAS function, whose authority is ±10° of deflection for the stabilator. As shown in Figure 1.2, the CAS function is applied by an additional electrical input on the mechanically signaled flight control actuator.

1.2.1.3. *Reduction of piloting forces*

When the forces to be applied on the driven load exceed human muscle capacity, a force amplification function has to be installed. Although some regional aircraft still use an aerodynamic assist in the form of servo tabs or control tabs, this amplification function is most often accomplished with actuators, referred to as primaries, which are nonetheless still overwhelmingly hydromechanical. Thus, primary flight control actuators (primary servo controls) are inserted in series between the pilot inceptors and the load to be moved, as close to the latter as the integration within the airframe allows. They are supplied by hydraulic power and they servo control the position of the driven load in response to the pilot demand, i.e. transmitted by the mechanical information chain. Therefore, these are hydromechanical actuators, which is to say, hydraulically powered and mechanically signaled, as shown in Figure 1.10. The power architecture and geometric integration of such a hydromechanical position servo actuator are discussed in Chapter 7 of Volume 1. For each critical function, the actuation is generally rendered redundant (triplex actuation, active–active–active, as shown in Figure 1.3, dual–duplex actuators, active–active, as shown in Figure 1.4) to achieve the required level of reliability.

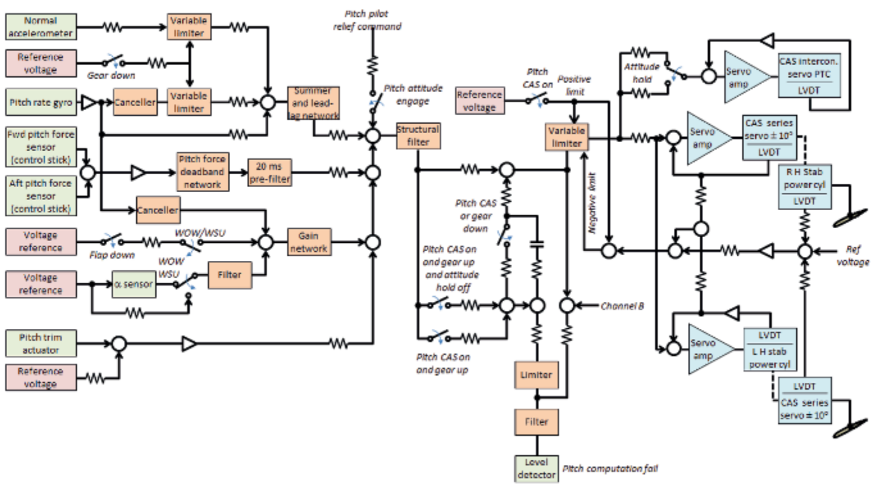

Figure 1.9. *Analog control diagram of the CAS servovalves for the pitch control of the McDonnel Douglas fighter F-15, (http://www.f15sim.com). For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

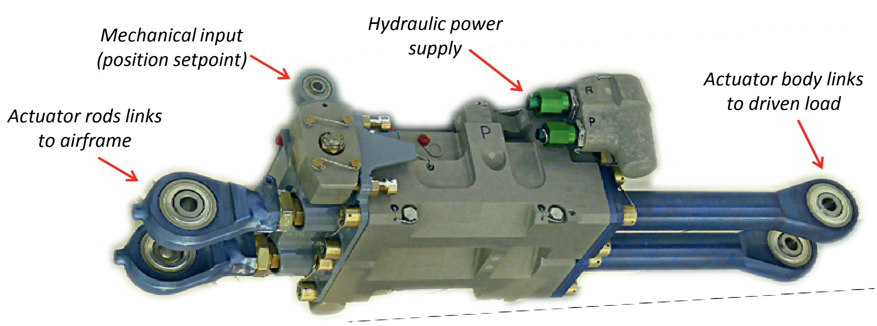

Figure 1.10. *Dual-parallel hydromechanical aileron actuator for the Falcon 900*

As the flight control actuators are irreversible, they do not reflect any force feedback to the pilot (to nearby forces due to unwanted effects in the mechanical signaling chain). It is therefore necessary to give the pilot a force feel[3] based on the position command which he sent to the actuators. For this, passive devices (artificial feel unit in Figure 1.3, pedal damper in Figure 1.4) create opposing forces on the mechanical information chain.

1.2.2. *Fly-by-Wire*

The transmission of information in mechanical form offers very good reliability, thereby often negating the need for the redundant path. Unfortunately, it has many flaws, whose importance often increases with the size of the aircraft and the number of loads to be moved. For example, the Boeing 747-400 has more than 20 flight control actuators some of which are located some 60 m away from the cockpit. It is thus clear that the effects of dilation, structural elasticity, friction or play in the transmission of command signals strongly penalize the accuracy, dynamics and geometric integration of the mechanical information chain. Moreover, the incremental evolution of adding auxiliary actuators on this command chain quickly reaches its limits in matters of authority, coordination of loads to be moved, mass, protection of the flight envelope and others. Airbus considers in particular that the introduction of FbW on its A320 has produced a real gain in mass of 200 kg. A similar gain was obtained by introducing the Load Alleviation Function (LAF).

As electrical signaling becomes more reliable, the more it becomes tempting to remove auxiliary actuators (AFCS, AP, SAS, CAS). It even becomes possible to completely remove the mechanical information chain if all of its functions can be achieved by signaling the primary actuators electrically. This leads to the Fly-by-Wire solution where the primary hydromechanical actuator is replaced by a Hydraulic Servo Actuator (or HSA)[4].

3 By his sense of touch or muscle reaction, the human being finds it easier to control forces rather than positions. In practice, this is often the mechanical impedance (relation force/position) of the actuated load that allows the human being to control positions. When this is not enough, a force feel system shall be introduced, for example through the installation of a simple spring.

4 An example for the power architecture of a hydraulic servo actuator for flight control can be seen in Figure 7.6 of Volume 1. Sometimes also called servo-hydraulic actuator or electro-hydraulic servo actuator.

1.2.2.1. *Full Fly-by-Wire*

Figure 1.11 is equivalent to Figure 1.7, when all the information is transmitted to/from the actuators in electrical form. We can note the complete disappearance of the mechanical information chain, the recourse sensors to measure the pilot's commands and the state of the actuator as well as the control and monitoring of the actuator mode (e.g. active/ disconnected/dampened/blocked). The aircraft feedback loop is shown to indicate, at least for monitoring, the position of the load to be actuated. Note that in practice, this position is somewhat different from the position provided by the actuator sensor. This comes from the kinematic gain variation of the actuator/load transmission, of the joints backlash, and of the deformation of solids under load.

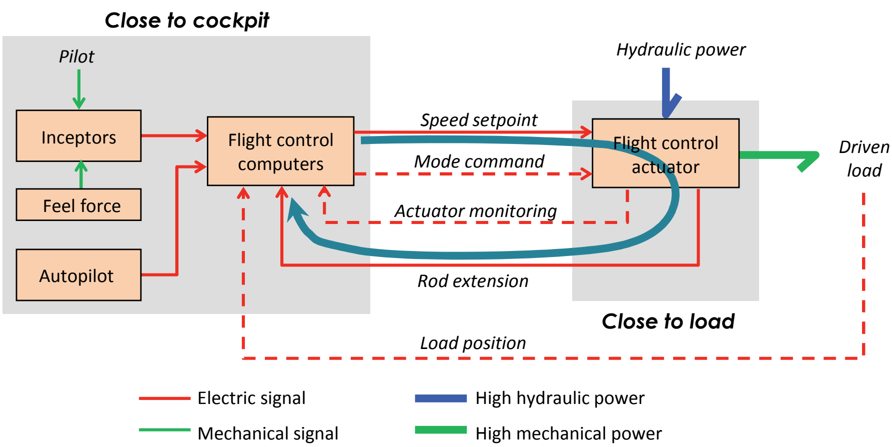

Figure 1.11. *Generic architecture, signal and power, of an electrically signaled flight control axis. For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

Figure 1.12 shows one of three full FbW actuators for the positioning of the rudder for the Boeing B777. It highlights interfaces for power (mechanical and hydraulic) and signals (electric). Internal power interfaces between electrical and mechanical domains (servovalve, mode selection solenoid valves or pressure reduction) are also displayed. The reaction and kick links (Figure 7.14 Volume 1) are not mounted.

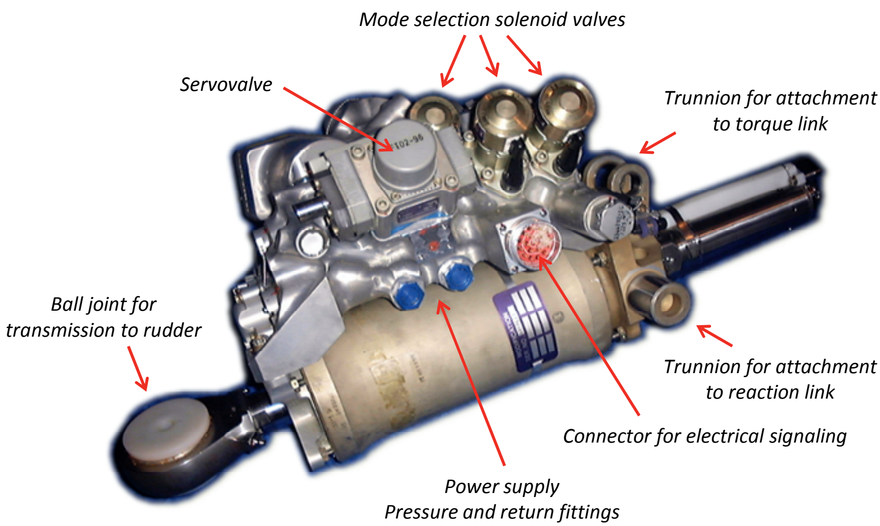

Figure 1.12. *Full FbW actuator for the rudder control of the Boeing B777*

The signal architecture shown in Figure 1.11 calls for several major remarks:

1) In conventional architectures, the primary actuator receives a position setpoint. It is in charge of locally performing, hydromechanically, the servo control position of the load according to this setpoint. This solution has the advantage of not being particularly vulnerable. However, it reduces the opportunities of advanced correction in the control loop [MAR 99]: overlapped/underlapped/dual slope valve or even the more complex Dynamic Pressure Feedback (DPF). On the one hand, in the FbW architecture of Figure 1.11, the actuator receives the servovalve current (a few mA) which is functionally representative of the desired load speed. Because the coils of the servovalve have a low impedance, another advantage of this solution is that it is not sensitive to noise or electromagnetic interference. Furthermore, as the power control of the servovalve (a few 10 mW) does not require high power electronics, its reliability is increased. In addition, the fact that the computers need to be located nearer the cockpit, in pressurized and temperature controlled avionic zones, helps to further increase reliability. However, when we consider that there may be several tens of meters between the computer and the actuators, it is understood that the transmission of commands or

sensor feedback is potentially vulnerable: depending on the type of fault, which can produce jamming, free or erratic movement, directly affecting the load to be actuated.

2) In a full FbW architecture, the interest in the removal of a number of mechanical components is offset by the number of electrical signals to be conveyed, typically between 15 and 25 wires per actuator channel (e.g. four wires to control the servovalve, three times two wires to control the mode solenoid valves, six wires for the ram position sensor, six wires for the position sensor of the mode selection valve). On a commercial aircraft, it is typically considered that there are between 350 and 450 electrical wires connecting the computers located in the cockpit to the flight control actuators [KUL 07]. FbW on a helicopter, such as the NH-90, for which electrical signaling is quadri-redundant, takes on average 60 wires for each of the four flight control actuators, totaling more than 200 electrical wires. Later on, it will be seen how the number and length of wiring has been reduced in recent programs.

1.2.2.2. *Pseudo Fly-by-Wire*

When the level of reliability achieved by full electric signaling is not enough[5], the solution is to double the electrical channel with a mechanical channel, hence introducing a dissimilarity that is beneficial for reliability. In the vast majority of cases, the electrical channel is the normal operation working chain and the mechanical channel is used as the backup. A counter example is the AH-64D Apache helicopter, in which the electric flight command is used as a backup in the event that functionality of the mechanical channel is lost. In this case, the pilot must land the helicopter as soon as possible because the reliability of the electric channel is not sufficient to continue the mission safely.

The pseudo FbW was implemented on the flight commands of the supersonic Concorde; for the yaw control and for the control of the trim horizontal stabilizer on the Airbus A320 and A330-200; to control the stabilizers of McDonnell F-18 fighter [HAR 83]; or as an intermediate step to full FbW for the NH90 helicopter, as shown in Figure 1.13. In this last example, the hydraulically powered primary actuator has both an electrical input and a mechanical input.

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5 For example, a failure rate lower than 10^{-9}/flight hour is typically required for a flight control axis (see Volume 1).

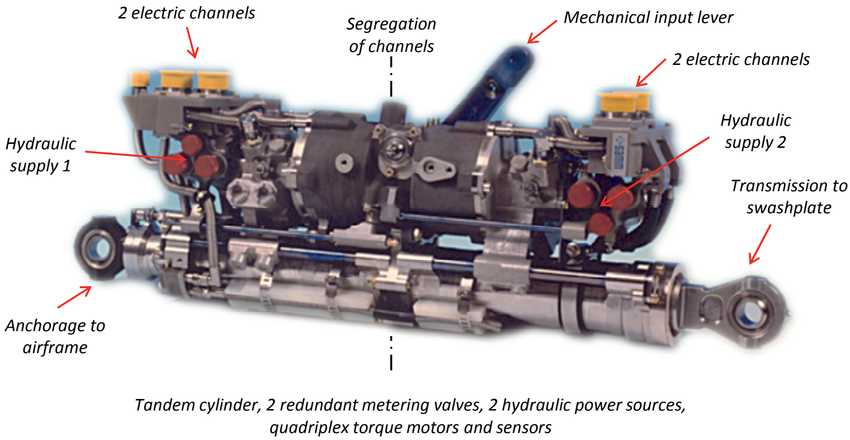

Figure 1.13. *Main rotor actuator of the NH90 helicopter, the pseudo FbW version used in the flight tests for development*

1.3. Challenges associated with electrical signaling

1.3.1. *Electrical interfaces*

Compared to conventional flight controls, the electrification of actuators imposes reliable interfaces to or from the electrical domain: power interfaces to meter or manage hydraulic power in the actuators from an electrical signal; human machine interfaces to collect the orders of the pilot; sensors to measure the physical quantities of the actuator and the aircraft.

1.3.1.1. *Power interfaces*

For hydraulically powered actuators, the interface between electric signaling and hydraulic power domains is performed by the servovalve that has been studied in detail in Volume 1, Chapter 5. From a very weak power of electric command (a few tens of mW), the servovalve meters high hydraulic power (tens of kW), giving it a gain in power in excess of a million. The power is metered with good dynamics (some 10 Hz bandwidth), good linearity (combined linearity/threshold/hysteresis of only a few % of the nominal range) and with an excellent capacity to reject external disturbances, in particular vibration and temperature.

The initial trend for military and space applications was to develop highly specific servovalves for each program. As such they could respectively accommodate internal mechanisms to stabilize the position control or satisfy the reliability requirements (power management and reconfiguration as a function of the desired response to a fault). The introduction of augmentation or autopilot functions created the dual input servovalves, as shown in Figure 1.14, of the Eurocopter EC225. The auxiliary electrical input issued by the functions of AP, AFCS or SCAS is superimposed on the primary mechanical input command issued by the pilot.

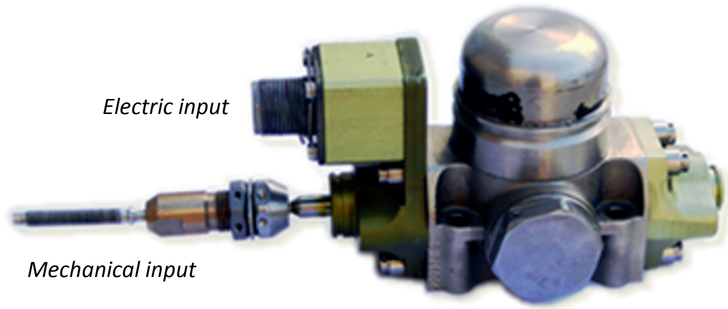

Figure 1.14. *Dual input servovalve for the SCAS of the Eurocopter EC225 helicopter*

1.3.1.2. *Interfaces to measure physical quantities within the actuator*

To overcome the great spatial extent of the electric position control loop in Figure 1.11 and to avoid recourse to an electric position sensor, the first FbW controls were implemented locally, that is to say at the level of the actuator, hence the position feedback was purely hydromechanical. Figure 1.15 shows this principle through the example of the Thrust Vector Control (TVC) of the main engines for the NASA Space Shuttle.

The electric signal input of the actuator thus consigned the position setpoint, which was locally transformed into electromagnetic force. The comparison function of the position servo loop was then realized by comparing this electromagnetic force to the elastic force produced by a spring cage as a function of the relative position between the actuator's rod and body. As of the 1960s, this type of solution is used to control the thrust vector of the space launcher programs for Gemini, Saturn and Space Shuttle, as well as on the SAS actuators of the General Dynamics F111 bomber.

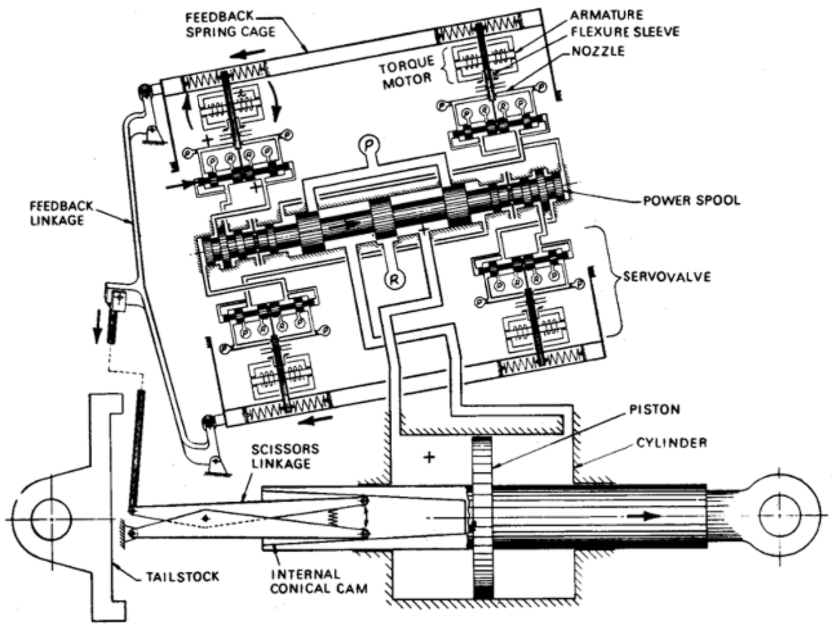

Figure 1.15. *Purely hydromechanical feedback loop actuator for the FbW TVC actuator of the main engine in the NASA Space Shuttle (© Moog Inc.)*

Later on, it came to implement the FbW position feedback electrically. Resistive potentiometric position sensors were discarded due to the presence of sliding contact. Friction impacted the lifespan (although the average position is quasi-invariant, the average speed is quite high), reduced reliability and generated measurement noise. Among the concepts for non-contact position sensors, the uptake of the variable differential transformer [MEA 13, NOV 99] has quickly become a standard feature for several reasons:

– it is well suited to the measurement of the absolute linear position, including for long strokes for this application; it is called Linear Variable Differential Transformer (LVDT). If necessary, it can also be extended for the measurement of absolute angular position, becoming a Rotary Variable Differential Transformer (RVDT). This is frequently used in the steering controls of auxiliary landing gear. As the measurement range of RVDTs is only a few tens of degrees, they can be combined if necessary with a backlash free mechanical gearbox;

– the sensor body that contains the coils can be easily sealed and made resistant to the operating pressure on the actuators (e.g. 350 bar);

– the core need not be guided precisely within the body: the diametral clearance may be several tenths of a mm;

– the ratiometric demodulation (see below) allows the sensitivity of the electrical output to temperature and voltage excitation to be greatly reduced.

An LVDT is a variable transformer. The body includes a primary excitation coil and two secondary measurement coils. The mechanical position to be measured is imposed by a non-magnetic rod to a ferromagnetic core. The displacement of the core relative to the body modifies the magnetic coupling between the primary coil and the two secondary coils, as shown in Figure 1.16.

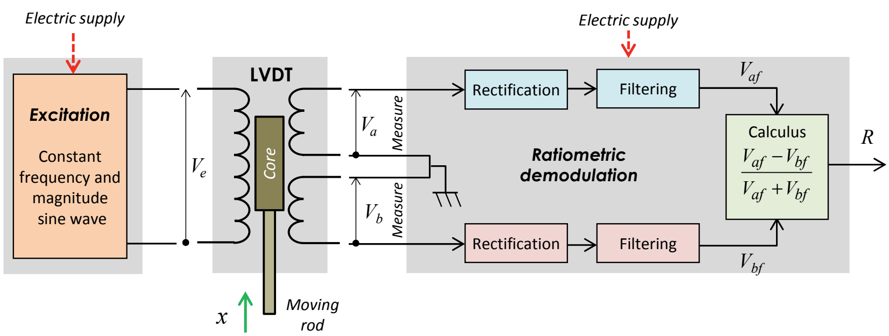

Figure 1.16. *Functioning principle of LVDTs and ratiometric demodulation*

The primary coil is fed by a sinusoidal voltage (a few kHz) of constant amplitude (a few V): the carrier. As a result, the current that arises generates a variable magnetic flux which induces voltages V_a and V_b in the two secondary coils. These voltages have the same frequency as the carrier but are amplitude modulated by the relative position of the core in regard to the body. To demodulate these voltages with a view to produce the electric signal that is representative of the position to be measured, the carrier must be eliminated by rectifying and low-pass filtering of voltages V_a and V_b. Two concepts are mainly implemented:

– the *synchronous demodulation* uses only the voltage difference V_a–V_b which is obtained by connecting the two secondary coils in series and in phase opposition. Thus limiting the number of wires of the sensor to four (two for excitation and two for measurement). To be precise and to demonstrate good linearity in the vicinity of the center position, this solution needs to compensate for phase shifts (introduced by the unwanted capacitances) between the voltages of the coils, which requires the use of active electronic circuitry.

– *ratiometric demodulation* is preferred in aviation because it eliminates the phase effects, the thermal drift and the influence of the excitation magnitude. To this end, the LVDT transducer is designed to keep the sum of the voltages V_a and V_b constant across the entire functional stroke. After rectification and filtering, these tensions can be directly processed by computer to form the ratio:

$$R = \frac{V_{af} - V_{bf}}{V_{af} + V_{bf}} \quad [1.1]$$

which is the electrical image of the core position relative to the LVDT body, a null ratio corresponding to the centered position. This type of demodulation has the disadvantage of using separate V_a and V_b voltages, which requires the electrical connection of six wires (three coils).

For their robustness to the environment and their accuracy, LVDTs are also used to measure other physical quantities where they can be turned into position: a force sensor by measuring the deformation under load; a pressure sensor by measuring the hydrostatic force on the surface of a membrane.

1.3.1.3. *Human–machine interfaces*

The removal of the mechanical chain transmitting the commands of the pilot offers new opportunities for the ergonomics of the cockpit. However, aircraft manufacturers have taken this benefit in different ways:

1) Central column vs. side-stick

The first option is to retain the control stick (or the central column with its roll control wheel) in line with conventional flight controls. In this way, the human–machine interface is unchanged for a pilot who progresses to a FbW aircraft. The second option is to introduce side-sticks in order to clear the space in front of the pilot. This improves vision from within a glass cockpit and/or allows for the possibility of touchscreens.

2) Passive vs. active central column or side-stick

The first option privileges reliability. It consists of using a passive column or side-stick, that is to say, only capable of opposing a predetermined force to the action applied by the pilot. Its main drawback is the inability to reproduce the mechanical link between the pilot and co-pilot commands. This feature enables one pilot to perceive the actions of the other and is nevertheless very useful both for training and also to detect instances of double pilotage. Equally, the removal of this feature makes it impossible for the pilot to feel, through the control stick, the commands generated by the autopilot or the vibrations generated by a shaker announcing the risk of stalling. Finally, the mechanical impedance of the control stick, that is to say, the force felt by the pilot in response to his actions, cannot be dynamically modified to suit the flight conditions and the specific ergonomic choices of the aircraft manufacturer.

The use of an active column or mini stick, which involves actuators capable of producing a driving force on the stick, remedies these disadvantages. However, in so doing it raises strong challenges for performance and reliability, especially in instances where side-sticks must be combined. An intermediate solution is to use a semi-active design. It only generates an opposing force, according to the desired mechanical impedance, that can be controlled electrically.

The supersonic Concorde was equipped with analog pseudo FbW on three axes. The control stick with the rolling wheel and the conventional rudder had been preserved. However, as the servo controls were irreversible, two force-controlled electrohydraulic actuators, shown in Figure 1.17, were associated in a parallel according to an active–passive architecture on each axis, in order to make the pilot feel an opposite force depending on flight conditions.

Before the introduction of FbW on its A320 in the mid-1980s, the European manufacturer, Airbus, had already conducted tests in 1978 on Concorde, and then again on a modified A300-B2 [ZIE 85]. The findings led them to retain the solution of a passive stick with position control (centimetric stroke) rather than force control. When the autopilot is engaged, the side-stick is blocked in the central position. In case of need, the pilot may override this blockage by applying sufficient force to disengage the autopilot and take manual control. In manual operation, normal flight control laws[6]

6 Other control laws are activated in case of faults: for example, alternative law (the protections are removed but the computers still apply correction in the actuators' position control loops) and direct law (the pilot's commands are transmitted unmodified to the flight control actuators).

consider action on the side-stick as a request for acceleration. The flight control computer has the authority to limit the request so as to keep the aircraft within its flight envelope: limiting load factor (e.g. between +2.5 and −1 G); attitude protection (e.g. a maximum of 30° for pitch-up and 15° for pitch-down); angle of attack; and overspeed protection.

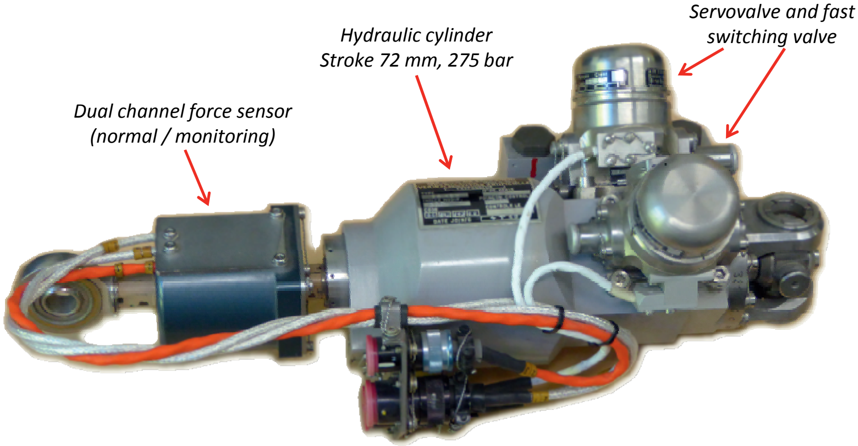

Figure 1.17. *Artificial feel actuator for the Concorde*

Since their introduction on the Airbus A320, the side-sticks, shown in Figure 1.18, have made significant progress in terms of integration [BET 15]. It is estimated that the choice of a side-stick over a central column saves 60 kg on the A320. The efforts and progress in integration thus save an estimated additional 5.5 kg on the A380, a further 3 kg on the A400M and another 2 kg on the A350 XWB.

Bell has chosen to expand the use of the side-stick for the 525 Relentless helicopter, which will be the first civil full FbW helicopter; with a side-stick on the left-hand for the cyclic controls and another side-stick on the right-hand for the collective pitch control.

For its part, Boeing chose to keep both conventional central columns, mechanically coupled, with the introduction of FbW on its B777. In this aircraft, the flight envelope remains under the complete authority of the human pilot: unlike the choice of the European aircraft manufacturer, commands are not limited by the flight control computers which only produce “dissuasive” limits through alerts and tightening of the control stick as the limits of the flight envelope are approached, so that the pilot forces the airplane to depart from its normal flight envelope.

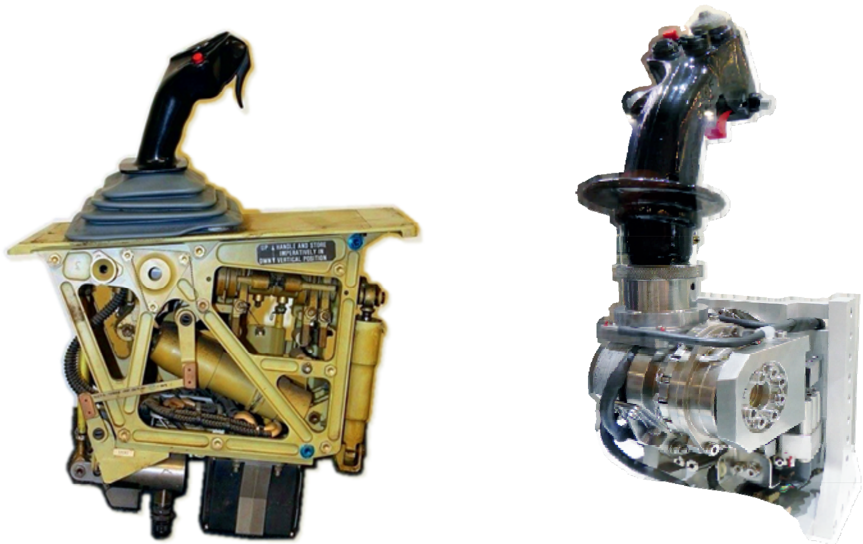

Figure 1.18. *Side-stick Left: passive (A320, courtesy of SwissTeknik LLC); Right: active (KAI T-50)*

Active side-sticks are already implemented for military applications, such as the Lockheed Martin F-35 multi-role fighter and the Sikorsky CH-53K heavy helicopter. According to [WAR 15], the feedback seems to have convinced aircraft manufacturers of the maturity of this option, which was selected for the Embraer KC390 medium-transport military aircraft and the Gulsftream G500/G600 business aircraft. As the certification requirements are much more stringent for civil than for military aircraft, side-sticks, should be, for example, dual–duplex (*dual* type for command/monitoring architecture on each channel and *duplex* for the presence of two channels associated in parallel), with two different processors and quadri-redundant sensors. Depending on the manufacturer's choice, both the control sticks of the pilot and the co-pilot can be connected "electronically" to replicate the existing conventional mechanical conjugation between the sticks and to reflect the actions of the autopilot.

1.3.2. *Evolution of the control and information transmission architectures*

As so often is the case with innovation, the reliability of FbW poses a dilemma for designers. First, greater use has to be made of the redundancies in order to meet the reliability requirements. Second, the multiplication of redundant channels needs to strike a balance with the mass and the increase in the complexity of information transmission, which in turn has an effect on overall reliability. It is a very gradual evolution that occurs over several years or even decades.

1.3.2.1. *Centralized analog FbW for actuator control*

At the beginning of the introduction for FbW (e.g. for the Concorde or the Dynamic General-F16), the actuator feedback loop was performed by analog electronic circuits situated in a less hostile environment, most often in the cockpit, in a pressurized and temperate zone.

1.3.2.2. *Centralized digital FbW for actuator control*

Analog circuits were then replaced by digital computers (e.g. Airbus A320), and still located with the avionics in the cockpit. In this architecture, both the issuing of the flight control commands and the closed-loop position control are performed far from the actuators. This is highly disadvantageous regarding the number and lengths of electric wires, as outlined above.

1.3.2.3. *Remote and mutualized digital FbW for actuator control*

The first step to achieve a more distributed architecture was taken in the mid-1990s, on the Northrop B-2 bomber [SCH 93] or the commercial aircraft Boeing B777 [YEH 96]. The actuator position control and monitoring functions are decentralized and integrated in electronic units; for example, eight Actuator Remote Terminals (or ART) for the B-2 or four Actuator Control Electronics (or ACE) for the B777. The ACE or ART interface with the digital avionics on the one hand (ARINC databus MIL-STD-1553 or ARINC 629, respectively) and with the analog electric signaling of the actuators on the other. There are still no electronics at the actuator level. As the position command is nearer to the actuators, the Flight Control Computers (or FCC) mainly transmit the actuator position commands in digital form. This is an advantageous solution for the required data rate. It is in fact much lower than would be necessary if the computers were having to "close" the position control loop and transmit the command to the actuator's servovalve.

Figure 1.19 shows a simplified diagram of the digital FbW architecture of the remote and integrated actuators' control for the Boeing B777.

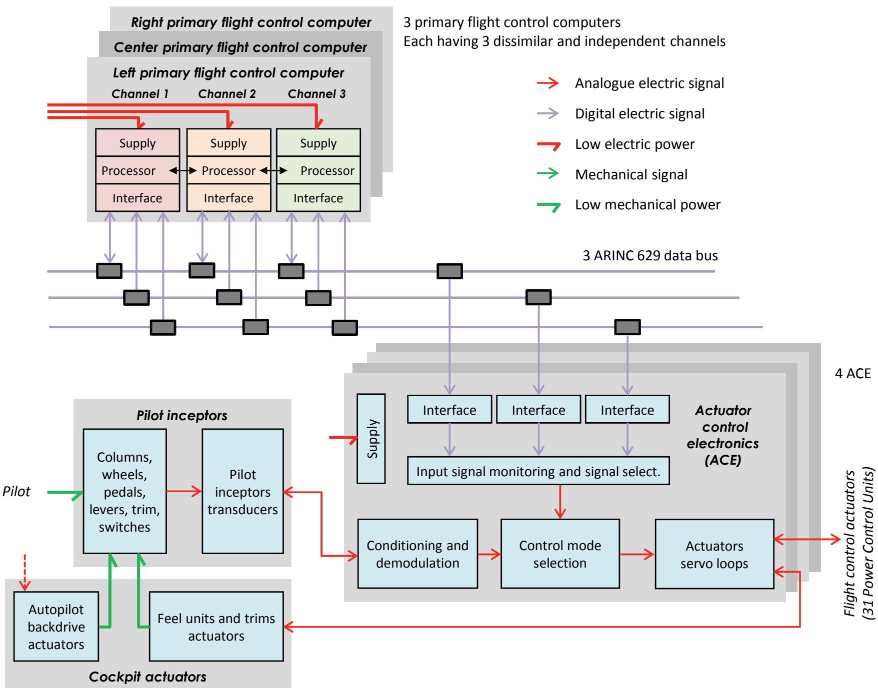

Figure 1.19. *Signal architecture with ACE in the Boeing B777. For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

1.3.2.4. *FbW with smart actuators*

The lessons learnt enabled more reliable installations of the electronics in harsher environments (temperature, pressure, humidity, vibration, electromagnetic interferences, lightning, high intensity magnetic fields). Therefore, it became possible to integrate the electronics within the flight control actuators to make so-called *smart actuators*.

One of the first applications for commercial aircraft concerns the trim horizontal stabilizer of the Airbus A330/A340, which integrated three digital electronic controls communicating, via the ARINC 429 databus (see section 2.2.3 of Chapter 2), with the flight control computers. On the A380, the electronic controls of the Electro-Hydrostatic Actuators (EHA) are used as backup (see Chapter5) and are designed to receive the same type of command signals as used by conventional hydraulic servo-actuators for the normal mode: a rated current of a few mA which is representative of the desired speed for the load to be moved. The integration of these electronics into the actuator for the local control of the motor (commutation, current and speed loops) was another step towards smart actuators. Yet another step was taken on the recent Boeing B787 and Airbus A350 XWB, as all flight control actuators, conventional or PbW, incorporated an electronic module (REU for Remote Electronic Unit or RAE for Remote Actuator Electronics). This module is responsible for carrying out the functions for position control and for the monitoring of the actuator. Equally, it also appears as an interface between the digital data on the side of the avionics databus and the analog electric signals on the side of the actuator (e.g. commands for control of power, signals returning from the sensors). The connection to the databus requires a very small number of wires (typically 4), compared with the number of wires (typically 15–25 depending on redundancies) that connect the actuator to this module [GOD 02].

1.3.3. *Reliability and backup channels*

To reach the levels of reliability required for FbW, it is usually necessary to install a backup channel. As shown in Tables 1.1 and 1.2, an intermediate solution, widely used in the electrification of the information chain, is to keep a mechanical channel as backup (e.g. the mechanical input on the actuator in Figure 1.13). The total removal of the mechanical linkages between the cockpit and the actuators imposes new constraints on matters of independence and dissimilarity for the entire backup chain (pilot inceptors, closed loop control, power sources for command).

1.3.3.1. *Backup with fluidic technology*

In the early 1970s, as part of the American research program FLASH (Fly-by-Light Advanced Systems Hardware), one of the concurrent goals was concerned with solutions towards fluidic computation [CYC 81]: the entire backup chain to transmit and process information between the pilot and the actuator was designed to be performed by pneumatic fluidics. This line of investigation was however quickly abandoned as the immunity of electrical solutions to electromagnetic interferences became significantly improved.

1.3.3.2. *Analog electrical backup*

One widely used solution consists of implementing the backup channel in analog electric technology. A good example is the evolution of the yaw control between the Airbus A340-200 and the A340–600 (see Figure 1.20).

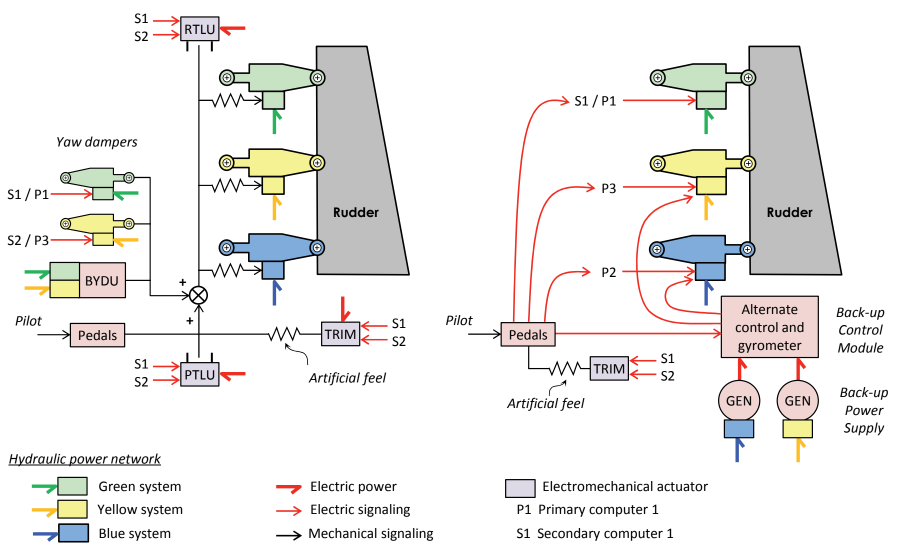

Figure 1.20. *Evolution of yaw control from hydromechanical actuation (Airbus A340-200, left) to electrohydraulic actuation (Airbus A340–600, right). For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*

On the -200 version, the actuation architecture was largely similar to that of the Airbus A320. This is presented in the left half of Figure 1.20:

– each of the three control actuators is mechanically signaled, hydraulically powered and locally realizes a closed-loop position control with a moving body design. The three actuators are associated with force-summing configuration and operate in the active–active–active mode to position the rudder;

– the mechanical commands defining the position setpoint of the three flight control actuators are identical. They are issued from the pilot or autopilot. The commands for the yaw damper and yaw trim that are produced at the tail of the aircraft are also added to this. Position commands are transmitted to the actuators through spring rods to ensure independence between channels in case one actuator fails;

– the mechanical yaw damper commands are generated by two electrically signaled hydraulic cylinders associated in force-summing configuration. As a backup, a Yaw Damper Backup Unit (BYDU) is added to the A320 design. It performs the yaw damping function in the event of a double failure of the yaw damper actuators (given that the A340 is more disposed to Dutch roll);

– to bound the aerodynamic structural loads, the position demand is limited according to the airspeed (Pedal Travel Limitation Unit or PTLU and Rudder Travel Limiting Unit or RTLU).

On the A340-600, the flexibility of the airframe required an increase in bandwidth and accuracy for yaw control. This led to the removal of the mechanical signaling channel in favor of full FbW. This architecture is shown at the right-hand side of Figure 1.20:

– the three fixed-body actuators are electrohydraulic, that is to say, hydraulically supplied and power metered by servovalve;

– besides the position closed-loop control function, the primary and secondary flight control computers are also in charge of the damping functions for yaw and travel limitation. They receive the position commands from the pilot in electric form through the pedal inceptors, and generate the control current for each actuator’s servovalve;

– the trim command remains electro-mechanical but is sent back to the cockpit;

– a backup channel is added to the yaw control. The constraints of autonomy, segregation and dissimilarity are cleared by means of an analog Backup Control Module (BCM) with its own gyroscopes, its own pedal sensors, as well as its own Backup Power Supply (BPS). This redundant supply produces electrical power through two generators, which are driven by hydraulic motors from centralized sources of pressure;

– a force equalization strategy between the three actuators, not indicated in the figure, is implemented in the position control of the actuators in order to limit force fighting.

1.3.3.3. *COM/MON architecture*

A major challenge in meeting reliability requirements is the ability to detect and isolate faults that may occur in the computation of the command within an actuation system, so that they do not result in flight control failure.

The COM/MON (Command/Monitoring) architecture briefly covered in Chapter 2 of Volume 1 and detailed in Figure 1.21, is a widely used solution [TRA 06]. The two channels COM and MON are simultaneously active. In terms of software they are completely independent, segregated and dissimilar. In principle, the command channel (COM) generates the commands for power metering and management that are transmitted to the actuator. The monitoring channel (MON) validates or inhibits their transmission to the actuator, and enables (or not), the active mode of the actuator (see Figure 7.6 of Volume 1).

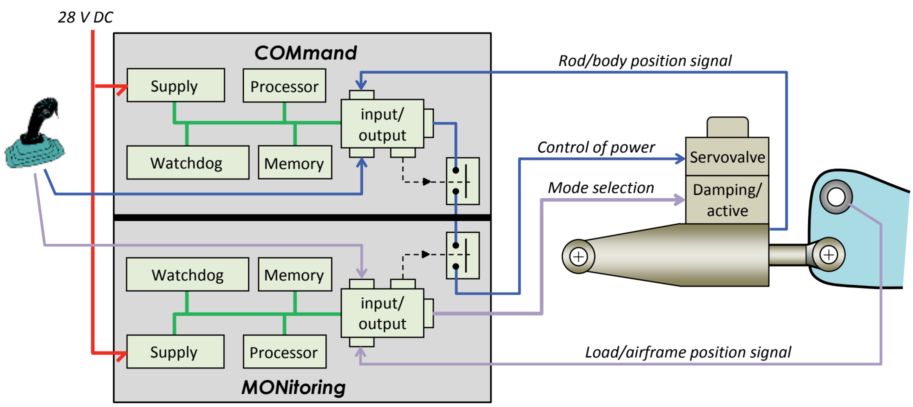

Figure 1.21. *COM/MON architecture for controlling the actuators (according to [TRA 06])*

1.4. The example of landing gears

The actuating functions related to landing gears (steering, braking, etc.) have broadly followed the same incremental changes of the flight controls. This development, similar to that observed in the automotive industry, can be illustrated through the example of braking. The left-side of Figure 1.22 shows a typical conventional braking system of an aircraft. For simplicity, a single circuit and a single wheel are represented and the parking brake is not mentioned. Each of the blocks is associated with different evolution steps:

– initially, the braking was purely physical, the forces being transmitted by a mechanical cable between the pilot and the wheels. Then the cables were replaced with hydrostatic power transmission, thereby overcoming the friction and compliance inherent to the previous transmission cables;

– the forces exerted by the pilot were assisted through the introduction of an external hydraulic power source;

– the anti-skid function was added to improve steerability and braking performance. It works by reducing the intensity of braking if excessive wheel/ground slippage is detected. Note that the anti-skid function, as shown in the figure as electronically controlled, appeared very early in purely hydromechanical form;

– finally, automatic braking (autobrake) helped to reduce the pilot workload for the management of the braking intensity.

It is clear that the assistance, anti-skid and autobrake features appear as add-ons on the original chain linking the pilot and the brakes. However, unlike flight controls, they are not located on the information chain, but rather the power chain, inserted as power metering functions in series (anti-skid) or in parallel (autobrake).

The right-hand section of Figure 1.22 shows the partial and simplified braking system of an airliner, taking full advantage of SbW (N.B. the redundant paths are not illustrated). The physical architecture is simplified and its integration facilitated. The control, brake assist and anti-skid functions report to a Braking and Steering Control Unit (or BSCU) and a single power metering device (pressure servovalve). The hydraulic lines between the cockpit and the braking system are replaced by electrical wires. A recent example of this illustrates well the scalability of SbW: combined with modular avionics architecture, the A380 or A350 Airbus can offer the *Brake to Vacate* function (or BTV). This feature aims to increase airport traffic by clearing the runway faster after landing. It is realized by automatically metering the braking of the wheels according to the position of the aircraft in relation to the ground in order to vacate the runway at a predetermined exit.

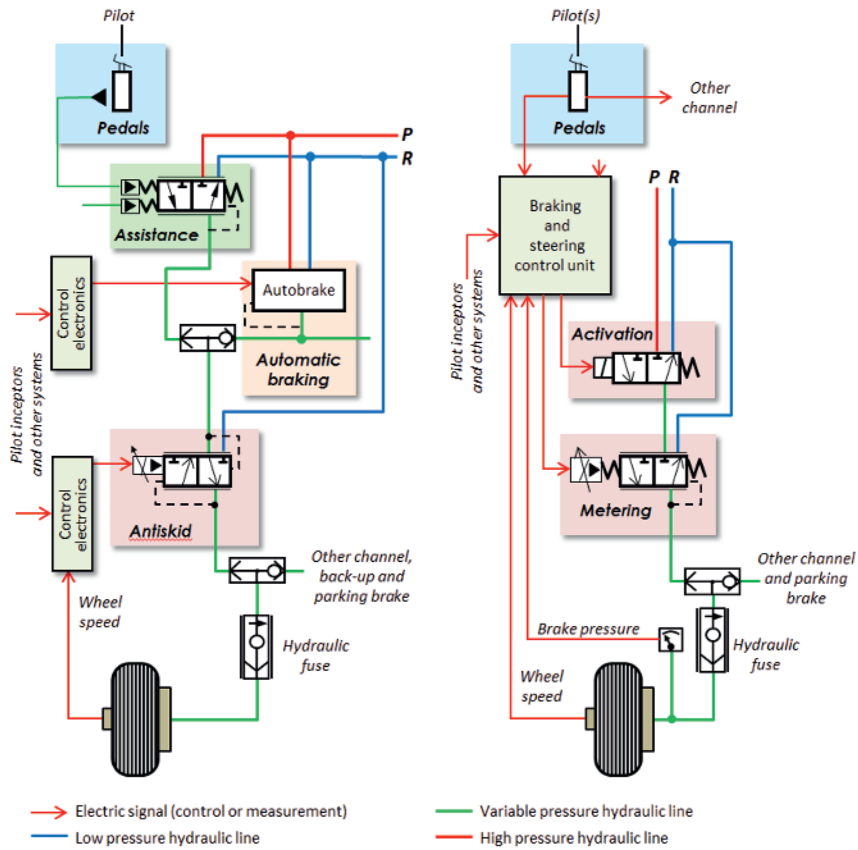

Figure 1.22. *Evolution of braking. Left: "conventional" brake with antiskid and autobrake (from [CRA 01]); right: fully integrated electrohydraulic brake. For a color version of this figure, see www.iste.co.uk/mare/aerospace2.zip*